REPORT DOCUMENTATION PAGE

1. REPORT DATE *(DD-MM-YYYY)*

2. REPORT TYPE
Technical Papers

3. DATES COVERED *(From - To)*

4. TITLE AND SUBTITLE

5a. CONTRACT NUMBER

5b. GRANT NUMBER

5c. PROGRAM ELEMENT NUMBER

6. AUTHOR(S)

5d. PROJECT NUMBER
1011

5e. TASK NUMBER
0062

5f. WORK UNIT NUMBER

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)

Air Force Research Laboratory (AFMC)
AFRL/PRS
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8. PERFORMING ORGANIZATION REPORT

9. SPONSORING / MONITORING AGENCY NAME(S) AND ADDRESS(ES)

Air Force Research Laboratory (AFMC)
AFRL/PRS
5 Pollux Drive
Edwards AFB CA 93524-7048

10. SPONSOR/MONITOR'S ACRONYM(S)

11. SPONSOR/MONITOR'S NUMBER(S)

12. DISTRIBUTION / AVAILABILITY STATEMENT

Approved for public release; distribution unlimited.

13. SUPPLEMENTARY NOTES

14. ABSTRACT

15. SUBJECT TERMS

16. SECURITY CLASSIFICATION OF:

a. REPORT	b. ABSTRACT	c. THIS PAGE
Unclassified	Unclassified	Unclassified

17. LIMITATION OF ABSTRACT

A

18. NUMBER OF PAGES

19a. NAME OF RESPONSIBLE PERSON
Leilani Richardson

19b. TELEPHONE NUMBER *(include area code)*
(661) 275-5015

6 Separate items are enclosed

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FROM: PROI (TI) (STINFO)   10 Jul 2000

SUBJECT: Authorization for Release of Technical Information, Control Number: **AFRL-PR-ED-TP-2000-147**
Keith McFall, "Solar Thermal – Solar Electric Propulsion Hybrid Orbit Transfer Analysis"

36ᵗʰ AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit   (Statement A)
(Huntsville, AL, 17-19 Jul 00)   **(Submission Deadline: 12 Jul 00)**

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SOLAR THERMAL – SOLAR ELECTRIC PROPULSION HYBRID ORBIT TRANSFER ANALYSIS

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20021118 083

36th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit
16-19 July 2000
Huntsville, Alabama

SOLAR THERMAL – SOLAR ELECTRIC PROPULSION HYBRID ORBIT TRANSFER ANALYSIS

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ABSTRACT

This effort examined the payoffs associated with the joint application of solar thermal propulsion (STP) and electric propulsion (EP) for orbit raising. The combined use of STP (800 second specific impulse) and EP (1800 second specific impulse) for a single orbit transfer mission is motivated by the desire to leverage the higher thrust of STP with the higher specific impulse of EP to maximize mission capability. The primary objectives of this analysis were to quantify the payload, mission duration, and hydrogen propellant to payload mass ratio for a range of combined STP and EP orbit transfer missions to geosynchronous Earth orbit (GEO), and contrast them to results for STP only. For STP, the hydrogen propellant to payload mass ratio is of particular interest due to payload fairing size constraints and the relatively low density of liquid hydrogen, which limit the mass of the STP propellant, and therefore the amount of payload that can be delivered. The results of the analysis include an 18% payload improvement associated with STP-EP hybrid propulsion over STP alone. The trip time needed for the STP-EP transfer varied from 101 to 143 days, compared to 41 days for the Solar only case. In addition, the amount of hydrogen propellant needed to accomplish the orbit raising to GEO per unit mass of payload decreased by 29% when the Solar Thermal – Solar Electric hybrid was used. While comprehensive comparisons of STP-EP to chemical propulsion (CP) only and to CP with EP orbit topping were also of interest, they were beyond the scope of this effort. However, a comparison of reference missions was performed. In comparison to the reference CP (328 second specific impulse) and CP-EP missions, the STP-EP system provided 67% and 39% payload increases, respectively. The trip time for the CP-EP cases varied from 55 to 106 days.

NOMENCLATURE

f_{tank} = tankage fraction

I_{sp} = specific impulse

m_{fract} = propulsion system stage mass fraction

m_{init} = initial stage mass

m_{pay} = payload mass

m_{prop} = propellant mass

δ_R = residual propellant fraction at conclusion of mission

ΔV = orbit transfer velocity change

INTRODUCTION

The Air Force Research Laboratory is supporting the development of Solar Thermal Propulsion technology under the Integrated High Payoff Rocket Propulsion Technology (IHPRPT) demonstrator and Solar Orbit Transfer Vehicle[1] programs. These efforts are motivated by the desire to increase the payload delivery capability of launch vehicle/orbit transfer systems while minimizing impacts with respect to orbit transfer time and spacecraft environmental exposure. STP systems for orbit transfer would use concentrated solar radiation to heat hydrogen propellant to high temperatures (>2500 °K), to reach specific impulses values of 800 seconds and greater. To generate the thrust level needed to accomplish low Earth orbit (LEO) to GEO transfer in 1-2 months requires the

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[*] Member AIAA

collection of 500-1000 kilowatts of solar power for the Delta IV M+ 5 meter fairing, 4 GEM-60 solid rocket motor (Delta IV M+ (5.4)) payload considered in this analysis. Lightweight, efficient inflatable concentrators[2], high performance heat exchangers to transfer collected power to the propellant, and long duration cryogenic hydrogen storage components and subsystems are currently being developed to support technology demonstration efforts for 50 kW class STP systems.

Numerous commercial and government organizations around the world are using EP systems[3-7] for on-orbit propulsion applications. Primex arcjets are being produced for the Lockheed Martin Corporation A2100 spacecraft. Russian Hall Effect Thruster (HET) systems have combined completed over 10,000 hours of on-orbit operation. the NASA NSTAR 30 cm ion engine is operating on Deep Space I, and Hughes 13 cm ion engines are on HS 601 HP spacecraft and 25 cm Ion Engines are on HS 702 spacecraft[3]. Development efforts for HET systems capable of supporting significant orbit raising are on-going under the USAF IHPRPT[7] program and NASA Advanced Space Transportation Program[6] (ASTP). Numerical investigations have reported significant potential increases in GEO spacecraft mass resulting from EP orbit raising. and solar array radiation degradation estimates have been performed[8,10].

Because commercial spacecraft bus power is current limited to 15+ kilowatts for GEO spacecraft, with 20+ kW class systems planned for next generation buses, the thrust of a STP system can be significantly greater than for an EP system. With the STP system beginning in LEO, relatively rapid orbit transfer through the inner Van Allen radiation belt can be achieved. Once above an altitude of approximately 15,000 km. the radiation dosage rate decreases significantly[9,11], and the higher specific impulse of EP can be used to complete the orbit transfer with commensurate improvements in payload lift capability. Application of EP for completing the transfer offers the additional advantage of reducing the hydrogen mass needed to complete the payload transfer. This is of particular importance due to the relatively low density of liquid hydrogen. and the launch vehicle fairing volume constraint.

ANALYSIS

Overview
The objectives of this analysis were to:
1) Quantify the payload. mission duration. and hydrogen propellant to payload mass ratio for combined STP and EP orbit transfer mission to GEO, and contrast them to results for STP only.
2) Compare the above result with CP and CP augmented with EP.

These objectives were accomplished using the following methodology:
1) Develop simplified numerical models of propulsion system performance.
2) Determine or assume launch vehicle and spacecraft propulsion system performance values.
3) Select orbit transfer missions and propulsion system stage architecture.
4) Calculate orbit transfer delta V and trip time.
- Model Solar Thermal Propulsion system orbit transfer.
- Incorporate the results of Electric Propulsion orbit transfer delta V and orbit transfer time calculated by Pollard[9] into the mass fraction model.
- Model chemical propulsion orbit transfer delta V assuming Hohmann transfer
5) Calculate GEO payloads. orbit transfer durations. and hydrogen mass usage for STP-EP hybrid, STP only, CP-EP hybrid. and CP only.

Propulsion System Performance
Solar Thermal Propulsion system: The STP system was modeled using specific impulse (I_{sp} : thrust averaged propellant exit velocity divided by 1 gravity acceleration), residual propellant fraction (δ_R : the fractional amount of propellant loaded into the propellant tank above that actually used to generate thrust), the propulsion system mass fraction (m_{fract} : the ratio between the loaded propellant and the wet mass of the STP upper stage), and the initial thrust to weight with respect to the entire spacecraft mass in LEO. Using the first three parameters, the amount of payload and loaded propellant can be determined as a function of mission delta V (ΔV : velocity change associated with orbital maneuvers). The fraction of the initial mass (m_{init}) that is payload (m_{pay}) and loaded propellant (m_{prop}) are given in equations (1.1) and (1.2) respectively.

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$$\frac{m_{pay}}{m_{init}} = 1 - \frac{\left(1 - e^{\frac{-\Delta V}{I_{sp}g}}\right)(1 + \delta_R)}{m_{fract}} \quad (1.1)$$

$$\frac{m_{prop}}{m_{init}} = \left(1 - e^{\frac{-\Delta V}{I_{sp}g}}\right)(1 + \delta_R) \quad (1.2)$$

Electric and Chemical Propulsion systems: The EP and CP systems were modeled using I_{sp} and tankage fraction (f_{tank}: tank mass divided by loaded propellant mass). Using these parameters, the ratio of payload to initial mass is given by equation (1.3). Note that each propulsion system will have its own distinct parameter values.

$$\frac{m_{pay}}{m_{init}} = 1 - \left(1 - e^{\frac{-\Delta V}{I_{sp}g}}\right)(1 + f_{tank}) \quad (1.3)$$

Propulsion system and Launch Vehicle Assumptions

Launch Vehicle: The Delta IV M+ (5,4) launch vehicle configuration is assumed for this analysis. The following characteristics and performance parameters are derived from reference 12. The assumed LEO (370 km circular orbit, 28.5 degrees inclination) separated spacecraft mass (the usable mass that separates from the launch vehicle) is 14217 kg (interpolated from 45 and 63.4 degree inclination data in Ref. 12)[13]. Separated spacecraft masses to highly elliptical orbits were linearly interpolated from Figure 2-27b of Ref. 12 and reproduced in the Appendix (Figure A-1).

Solar Thermal Propulsion System: The specific impulse, residual propellant fraction, STP system upper stage mass fraction, and initial thrust to weight were fixed at 800 seconds, 0.06, 0.7, and 5×10^{-4}, respectively. These performance characteristics were chosen to be consistent with IHPRPT program development efforts.

Electric Propulsion System: The specific impulse, and initial satellite acceleration under EP propulsion are needed to incorporate the EP orbit transfer analysis results of Pollard[9] into this work. In Pollard's analysis, values of 1600 seconds, and 3×10^{-4} m/s^2 were used. The specific impulse represented state of the art 1.5 kW Hall propulsion technology at the time. In the analysis performed for this paper, the values selected were 1800 sec and 3×10^{-4} m/s^2. To exactly match the acceleration

profile of Pollard's paper throughout the orbit transfer, the specific impulse values would need to be identical. However, the acceleration difference induced by the different specific impulse values is negligible (~ 1% at the end of the orbit transfer) for even the highest EP delta V case considered (1820 m/s). In summary, the propellant mass used is accounted for exactly and the spacecraft trip time will be slightly longer (<<1%) than projected in this analysis.

The EP tankage fraction and overall propulsive efficiency are also used to determine the final GEO payload (Eq.(1.3)) and beginning of life (BOL) specific power (power/unit wet mass). A tankage value of 0.1 was used based on previous published analyses[8,10]. An overall propulsive efficiency of 51.7% (based on a thruster efficiency of 55% and power processing unit efficiency of 94%)[3] was used to determine the BOL specific power for the required initial acceleration. This resulted in a BOL specific power of 5.1 w/kg (slightly lower than the value of 6 w/kg used in Ref. 9 due to different efficiency and specific impulse values selected).

Chemical Propulsion System: CP system performance was defined by tankage fraction (0.08) and specific impulse (328 seconds). Both values were based on SOA chemical bipropellant systems used in previous analyses[8,10].

Orbit Transfer Missions

Four different orbit transfer scenarios were investigated in this analysis.

1) In the first, termed the Solar + Electric mission, the launch vehicle injects into a 370 km circular orbit (28.5 degree inclination) a solar thermal upper stage mated to the satellite payload with an integrated electric propulsion system on-board. The solar thermal stage performs perigee burns (for relatively long burn arcs about perigee) to reach a high apogee. In next phase, apogee burns are made to raise the orbit perigee and modify the orbit inclination. Once the STP system has reached the desired EP starting orbit, it separates from the satellite payload. The satellite's on-board EP system is then used to complete the orbit transfer to GEO (0 degrees inclination). The starting EP orbits are constrained for this analysis to those calculated in reference 9. From that data, a subset was selected for use in this analysis. This set had the perigee fixed at 15,000 km altitude, apogees of 37000, 49000, and 61000 km altitude, and inclinations of 0, 4, 8, 12, 16, and 20 degrees.

2) In the second mission scenario, termed the Solar Only mission, the launch vehicle injects into a 370 km altitude circular orbit (28.5 degree inclination) a solar

thermal upper stage mated to the satellite payload that provides no EP for orbit transfer. This is a standard mission proposed for STP. The solar thermal stage performs perigee burns (for relatively long burn arcs about perigee) to reach an apogee near GEO. In next phase, apogee burns are made to raise the perigee and perform inclination changes to reach GEO (0 degrees inclination).

3) In the third scenario, the Chemical + Electric mission, the launch vehicle was used to inject the satellite payload with integrated electric and chemical propulsion systems on-board into highly elliptic geotransfer orbits (GTO). These orbits had a perigee of 185 km altitude, apogees of 37000, 49000, and 61000 km altitude, and an inclination of 27 degrees. The CP system is used to change the orbit to the EP starting orbits given above for the Solar + Electric case. The EP system completes the orbit transfer to GEO (0 degrees inclination). The EP analysis results from reference 9 were used in these calculations. This scenario was examined to provide a reference for the Solar + Electric case for the specific assumptions included in this paper.

4) In the forth scenario, the Chemical Only mission, the launch vehicle was used to inject the satellite payload with an integrated chemical propulsion system for orbit transfer into a GTO orbit (185 km perigee altitude, 35790 km apogee altitude, 27 degrees inclination). The CP system is used to raise to perigee to 35790 km altitude and reduce the orbit inclination to zero. This scenario was also examined as a reference for the Solar + Electric case.

Orbit Transfer Analysis

Solar Thermal Orbit Transfer: The orbit transfer calculations for STP involved the solution of 6 coupled equations of motion for orbit transfer and one equation associated with the mass change of the vehicle. Since the final orbit, GEO, had eccentricity and inclination values both equal to zero, the equations of motion were represented in equinoctial form[14] to remove calculation singularities. The numerical differential equation integrator in Mathematica[15] (version 4.0.0.0) was used to evaluate the orbital elements throughout the orbit transfer. No orbital perturbations, other than spacecraft thrust, were included. No eclipse effects were included.

The calculation method divided the orbit transfer into orbital cycles, each consisting of approximately 360 degrees along the orbit angular path. The variable of integration was the true anomaly based on the previous cycle's ending orbital elements. At the conclusion of each cycle, the variable of integration (a pseudo true anomaly)

was updated to reflect changes induced by propulsion system thrust.

For this analysis, perigee burns were made from –60 to +60 degrees about the pseudo periapsis (pseudo true anomaly equal to zero degrees). No inclination change was introduced during perigee burns. The thrust vector was along the vehicle's velocity vector. Apogee burns were made from –30 to +30 degrees about the pseudo apoapsis (pseudo true anomaly equal to 180 degrees). A constant out of plane thrust angle was used to induce the inclination change. The in-plane component of the thrust vector was along the vehicle's velocity vector.

The calculation model does not include a control loop to conclude the orbit transfer at the exact final orbit desired. Instead, the program computes the Hohmann delta V deviation between the current orbit and the desired final orbit. It then outputs the projected remaining delta V equal to the Hohmann deviation multiplied by the ratio of the current phase's (either apogee or perigee raising phase) delta V divided by the Hohmann delta V for the same transfer. Since the remaining delta V is typically under 100 m/s (and never over 200m/s) a delta V error of less than 20 m/s is anticipated. The transfer time is scaled in a similar manner, with errors less than 1 day anticipated.

The final outputs for each orbit transfer were the delta V and transfer time.

Electric Orbit Transfer: The electric orbit transfer results were taken directly from the work by Pollard[9]. A subset of those results is shown in figures A-2 and A-3.

The final outputs for each orbit transfer were the delta V and transfer time.

Chemical Orbit Transfer: The chemical transfer was calculated using the separated spacecraft masses intepolated from data in reference 12 (data is shown in figure A-1), and calculated delta V values to reach the starting EP orbit or GEO for the Chemical Only case. The chemical delta V (figure A-4) was computed assuming a Hohmann delta V using basic orbit transfer equations[16].

The final outputs for each orbit transfer was the delta V. The CP transfer time, negligible compared to Solar and Electric values, was taken to be zero.

RESULTS

Solar Thermal – Solar Electric Hybrid Propulsion
The solar thermal orbit transfer delta V and transfer time for LEO (370 km circular altitude, 28.5 degrees inclination) to the following orbits were calculated:

1) GEO (35790 km circular altitude, 0 degree inclination): Solar Only
2) Highly Elliptical Orbits (HEO): Solar + Electric
- 15000 km perigee altitude, 37000 km apogee altitude, orbital inclinations of 0,4,8,12,16, and 20 degrees
- 15000 km perigee altitude, 49000 km apogee altitude, orbital inclinations of 0,4,8,12,16, and 20 degrees
- 15000 km perigee altitude, 61000 km apogee altitude, orbital inclinations of 0,4,8,12,16, and 20 degrees

The delta V and transfer time results are shown in figures 1 and 2, respectively. Results in figure 1 show a significant reduction in Solar Thermal delta V, up to 22% for the 37000 km apogee altitude (20 degree inclination) case, as the EP system increases its contribution to the orbit transfer. The concentration of data points for the 8 degree inclination case is believed to be a random occurrence resulting from the sub-optimal selection of orbit transfer inputs (apogee burnout altitude, out of plane burn angle, etc.). The trip time results shown in figure 2 show a negligible variation in transfer time as a function of final inclination. The transfer times for the 37000 km altitude apogee cases are aproximately 19% less than for the 61000 km cases. The Solar Only trip time is not notably different from the 37000 km and 49000 km cases.
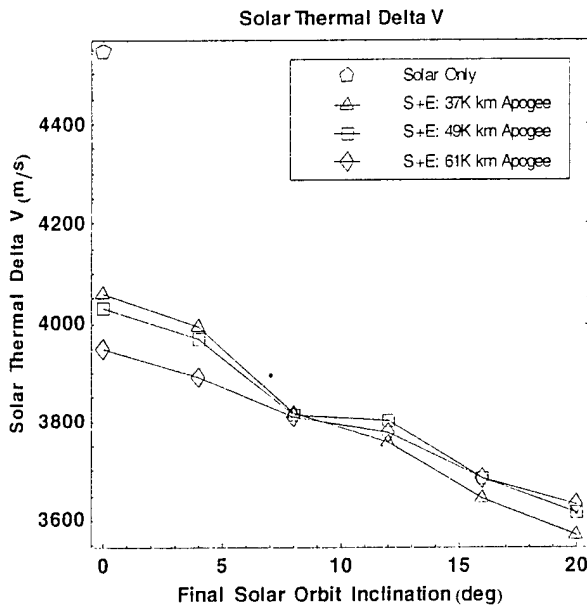

Figure 1.   Calculated Solar Thermal delta V from 370 km circular orbit (28.5 degrees inclination) to the final Solar Thermal orbit. For the Solar Only case, the final orbit is geosynchronous (0 degrees inclination). Final apogees and inclinations are shown on the plot.

Combining the Solar Thermal transfer results obtained in this study with the EP results from Pollard[9], the benefits of Solar Electric augmentation to Solar Thermal propulsion become apparent (Fig. 3). The potential payload lift capability increases by up to 18% over the Solar Only

case. Though not noted in figure 3, the payload and transfer time increase as the EP starting inclination increases from 0 to 20 degrees. The transfer time is seen to increase from approximately 40 days for Solar Only to between 120 and 140 days for the highest payload Solar + Electric cases.
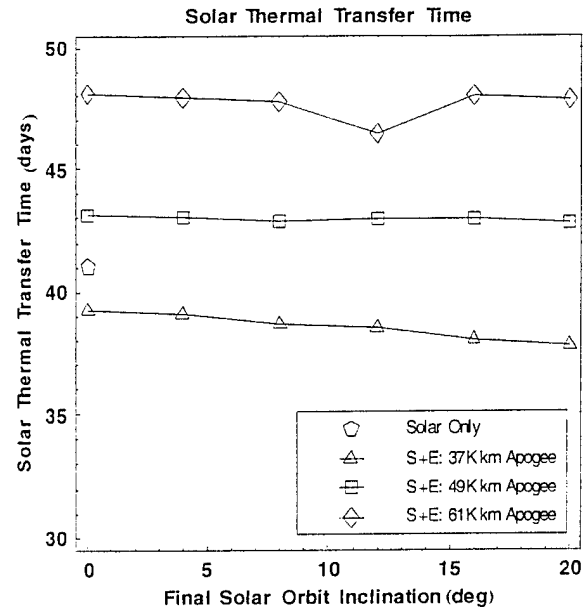

Figure 2.   Calculated Solar Thermal transfer time from 370 km circular orbit (28.5 degrees inclination) to the final Solar Thermal orbit. For the Solar Only case the final orbit is geosynchronous (0 degrees inclination). Final apogees and inclinations are shown on the plot.
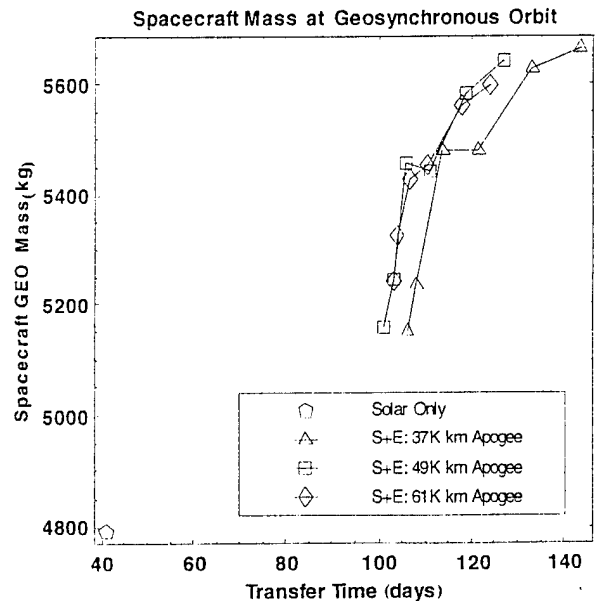

Figure 3.   Calculated spacecraft payload mass versus transfer time to GEO for Solar Thermal + Electric Propulsion and Solar Thermal only. Includes solar electric mission delta-V and transfer time results by Pollard[9]

The hydrogen mass required for raising a unit mass of spacecraft payload is shown in figure 4. A decrease of up to 29% when STP is augmented with EP is indicated for the cases examined. This means that for missions constrained by the volume of liquid hydrogen that can be contained within the launch vehicle fairing, EP can enable up to a 29% increase in payload mass to orbit.
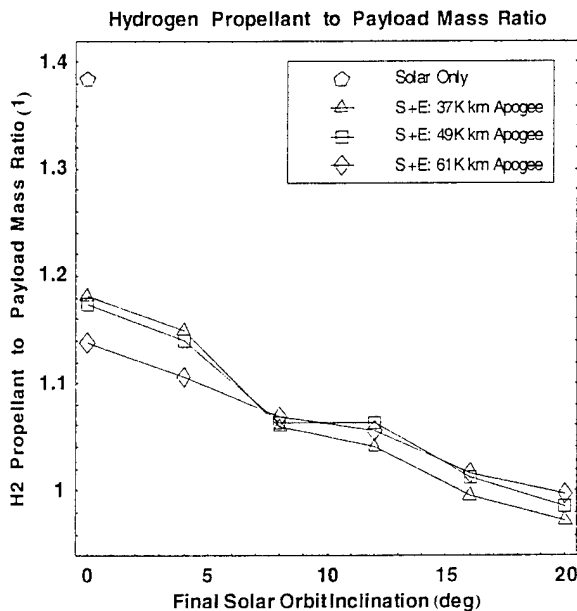
Hydrogen Propellant to Payload Mass Ratio

Figure 4.   Calculated Solar Thermal –Solar Electric Hybrid H2 propellant to payload mass ratio for transfers from 370 km circular orbit (28.5 degrees inclination) to geosynchronous orbit (0 degrees inclination). Includes solar electric mission delta-V and transfer time results by Pollard[9]

With respect to liquid hydrogen volume, a simple estimation of tank volume was performed to identify whether hydrogen volume is an issue for the Solar + Electric case. Of the cases investigated, the minimum specific hydrogen mass (hydrogen to payload mass ratio) is for the 37000 km apogee altitude, 20 degree inclination case. For this case, the spacecraft GEO payload is 5660 kg and hydrogen mass is 5510 kg. Assuming that all of the hydrogen in the tank is liquid, a best case assumption for propellant density is approximately 70 kg/m³ (Ref. 17). Assuming a cylindrical tank with an internal diameter of 4 meters (to fit within the Delta IV M+ (5,4) internal fairing diameter of 4.572 m)[12] a propellant tank cylinder length of approximately 6.3 meters would result. Assuming a payload adapter fair (PAF) height of 2.2 meters (consistent with the examples given in Ref. 12), the cylindrical tank would fill the entire constant diameter section of the 14.3m long fairing, leaving the upper conical section for payload. In reality, the hydrogen density would be less due to non-liquid hydrogen, and rounded end caps would be used on the propellant tank. On the other hand, lengthening the payload fairing to approach that of the

19.1 m Delta IV Heavy[13] fairing, and increasing the specific impulse of the STP system beyond 800 seconds would be considered to reduce hydrogen mass limitations. However, this simple volumetric analysis demonstrates the importance of reducing the amount of hydrogen needed to raise the spacecraft payload.

Solar Thermal and Chemical System Results

A comparison between the payload lift potential of STP and CP is presented in figure 5. In comparison to the reference CP and CP-EP missions, the STP-EP system also compared favorably, providing 67% and 39% payload increases, respectively. The trip time for the CP-EP cases varied from 55 to 106 days.

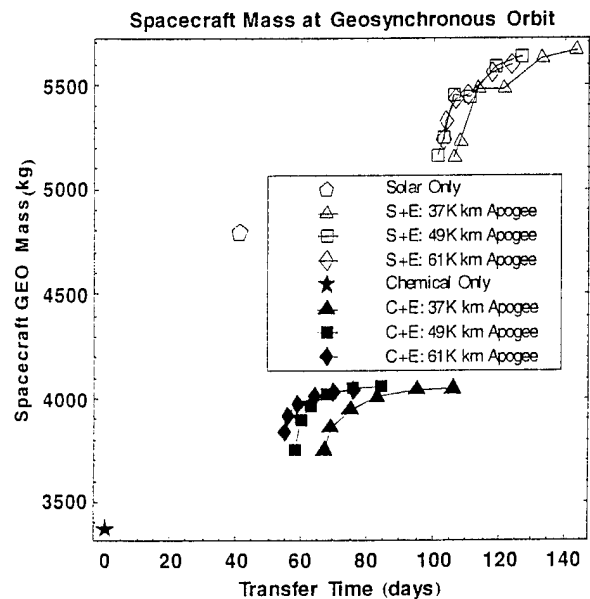**Spacecraft Mass at Geosynchronous Orbit**

Figure 5.   Calculated spacecraft payload mass versus transfer time to GEO for Solar Thermal + Electric Propulsion, Solar Thermal Only, Chemical + Electric Propulsion, and Chemical Only. Includes electric propulsion mission delta-V and transfer time results by Pollard[9].

It again must be noted that detailed launch vehicle payload fairing volume constraints have not been included in the analysis. In addition, the electric propulsion orbit transfer missions were constrained to only a few cases.

DISCUSSION

As was stated previously, the potential payload increases resulting from the application of EP for orbit raising with CP have been well demonstrated. Comparable improvements for STP-EP over STP only were anticipated and demonstrated in this effort. Of significant importance is the potential to greatly reduce the hydrogen propellant mass associated with payload orbit raising. EP was shown to offer significant reductions in this parameter. Increasing

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the specific impulse beyond the relatively conservative 800 seconds will also offer propellant reductions and is expected to be a focus of future AFRL STP propulsion development efforts.

Application of orbit optimization techniques such as those used for low thrust propulsion orbit transfer[8,10,18-20] is needed to optimize the joint application of solar thermal and solar electric propulsion.

CONCLUSIONS

The results of the analysis demonstrate the significant improvements in payload (18%) and hydrogen propellant usage (29%) that can be enabled through the joint application of Solar Thermal and Solar Electric hybrid propulsion when compared to Solar Thermal propulsion alone. Because this study focused on a limited set of missions, further analysis of hybrid transfer missions is needed to maximize payload and minimize hydrogen usage.

Because of scope limitations for the analysis, the comparison between STP-EP and CP-EP should be considered viable for the specific cases considered, not as an indication of the best that either hybrid system is capable of. With that caveat, in comparison to reference CP and CP-EP missions, the STP-EP system also compared favorably, providing 67% and 39% payload increases, respectively.

ACKNOWLEDGEMENTS

The author acknowledges the support of Mr. David Perkins and Dr. Michael Holmes.

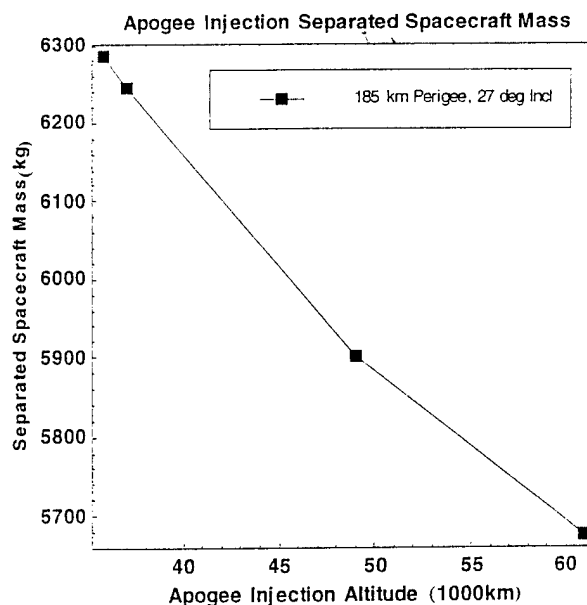## APPENDIX

Apogee Injection Separated Spacecraft Mass

Figure A-1. Separated Spacecraft Mass for chemical propulsion tranfer from orbit of spacecraft separation from Launch Vehicle (Delta IV M+ (5.4)). Masses were interpolated from the table on Figure 2-27a of Ref. 12. For the Chemical Only case, on-board chemical propulsion is used to transfer from geosynchronous transfer orbit (185 km x 35790 km alt, 27 degree inclination) to GEO.

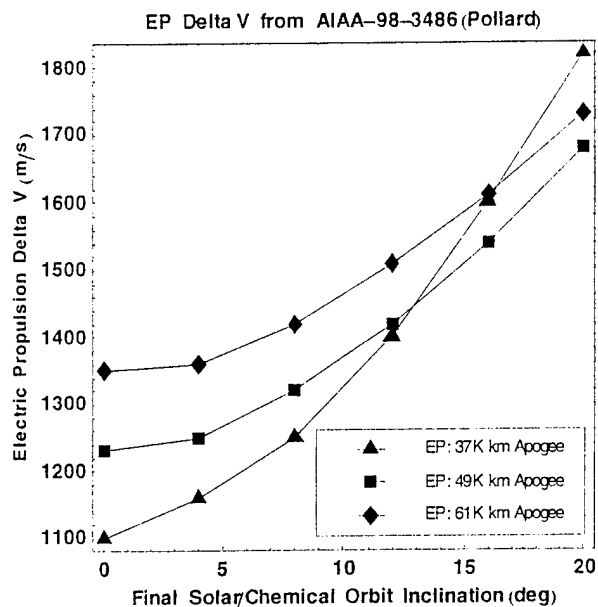

EP Delta V from AIAA-98-3486 (Pollard)

Figure A-2. Solar Electric delta V from 15000 km perigee (apogee and inclination shown on plot) to geosynchronous orbit (0 degrees inclination) as shown in AIAA paper 98-3486 (Ref. 9). Final apogees and inclinations are shown on the plot. The payload's Solar Electric propulsion system was stated to provide an initial acceleration and specific impulse of 3×10^{-4} m/s^2 and 1600 sec, respectively.

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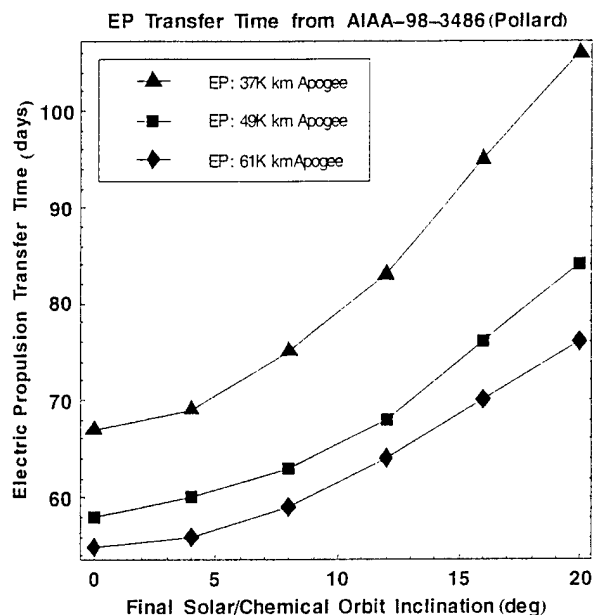**EP Transfer Time from AIAA-98-3486 (Pollard)**

Figure A-3. Solar Electric orbit transfer time from 15000 km perigee (apogee and inclination shown on plot) to geosynchronous orbit (0 degrees inclination) as shown in AIAA paper 98-3486 (Ref. 9). Final apogees and inclinations are shown on the plot. The payload's Solar Electric propulsion system was stated to provide an initial acceleration and specific impulse of 3×10^{-4} m/s² and 1600 sec, respectively.
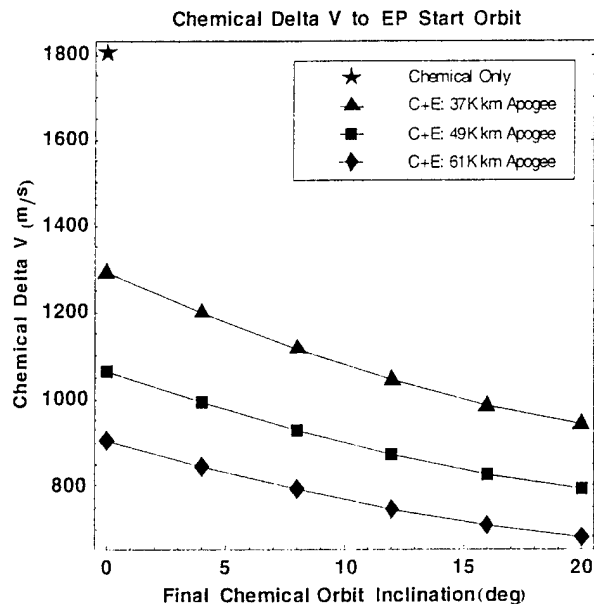
Chemical Delta V to EP Start Orbit

Figure A-4. Calculated Hohmann transfer delta V for chemical propulsion tranfer from orbit of spacecraft separation from Launch Vehicle. For the Chemical Only case, the spacecraft's on-board chemical propulsion system is used to transfer from geosynchronous transfer orbit (185 km x 35790 km alt, 27 degree inclination) to GEO.

REFERENCES

1. Partch. R.. Frye P., "Solar Orbit Transfer Vehicle Space Experiment Conceptual Design", AIAA paper 99-2476, 35th Joint Propulsion Conference. Los Angeles. CA. June 1999.

2. Paxton. J.. Hawk, C., "Material Property Effects on a Thin Film Solar Concentrator for Solar Thermal Propulsion", AIAA paper 94-3029, 30th Joint Propulsion Conference. Indianapolis, IN. June 1994.

3. Curran, F.M., "Electric Propulsion Activities in U.S. Industry", IEPC-99-001, Kitakushu, Japan, October 1999.

4. Garkusha, V., Lukiaschenko, V., Semenkin, A., Tverdohlev, S., Kim, V., Popov, G., *Maslennikov, N.*, Murashko, V., "Modern Status of Hall Thrusters Development in Russia", AIAA paper 99-2157. 35th Joint Propulsion Conference. Los Angeles, CA. June 1999.

5. Tahara. H., Nishida. M., "Overview of Electric Propulsion Activity in Japan", AIAA paper 99-2159, 35th Joint Propulsion Conference, Los Angeles. CA. June 1999.

6. Dunning. J., Sankovic, J., "NASA's Electric Propulsion Program", AIAA-99-2161, 35th Joint Propulsion Conference. Los Angeles, CA. June 1999.

7. Spores, R.A.. Birkan, M, "The USAF Electric Propulsion Program", IEPC-99-005, Kitakushu, Japan. October 1999.

8. Oleson. S.R. Myers. R.M. "Launch Vehicle and Power Level Impacts on Electric GEO Insertion", AIAA paper 96-2978, 32th Joint Propulsion Conference, Lake Buena Vista. FL. July 1996.

9. Pollard, J.E., "Evaluation of Low-Thrust Orbital Manuevers", AIAA paper 98-3486. 34th Joint Propulsion Conference, Cleveland. OH. July 1998.

10. Oleson. S.R.. "Advanced Electric Propulsion for RLV Launched Geosynchronous Spacecraft", IEPC paper 99-185. Kitakushu. Japan. October 1999.

11. Hastings. D.H. Garrett. H.. Spacecraft Environment Interactions, Cambridge University Press, NY, 1996, pg. 73.

12. Delta IV Payload Planners Guide, The Boeing Company. Huntington Beach. CA, October 1999.

13. Perkins. D.. Private Communications, AFRL/PRST. Edwards AFB, CA.

14. Battin. R.H., An Introduction to the Mathematics and Methods of Astrodynamics, AIAA, NY, 1987, pp. 490-494.

15. Wolfram. S., The Mathematica Book, Cambridge University Press, NY, 1996.

16. Bate. R.R, Mueller, D.D. White, J.E.. Fundementals of Astrodynamics, Dover Publications, NY, 1971. pp. 162-173.

8

17. Weast, R.C., <u>CRC Handbook of Chemistry and Physics</u>, CRC Press, FL, 1977, pg. B-25.
18. Betts, J.T. "Using Sparse Nonlinear Programming to Compute Low Thrust Orbit Transfers", The Journal of Astronautical Sciences, 41, 1993, pp. 349-371.
19. Scheel, W.A., Conway, B.A., "Optimization of Very-Low-Thrust, Many-Revolution Spacecraft Trajectories", AIAA Journal of Guidance, Control, and Dynamics, Vol. 17, No. 6, November-December 1994.
20. Enright, P.J., Conway, B.A., "Optimal Finit-Thrust Spacecraft Trajectories Using Collocation and Nonlinear Programming", AIAA Journal of Guidance, Vol. 14, No. 4, September-October 1991.